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NASA PROJECT APOLLO WORKING PAPER NO. 1014

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# PRELIMINARY CONCEPT OF APOLLO SPACECRAFT COMMUNICATIONS AND TRACKING EQUIPMENT

[redacted]

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

SPACE TASK GROUP

Langley Field, Va.

March 20, 1961

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# PRELIMINARY CONCEPT OF APOLLO SPACECRAFT COMMUNICATIONS AND TRACKING EQUIPMENT

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## SUMMARY

The Apollo vehicle communications and tracking system can be developed to the desired level within the designated time period by detailed design studies supported by the information and experience gained from Project Mercury and other related space programs. In the proposed system, first priority is given to that equipment necessary to maintain life in space and to bring the crew and vehicle safely back to earth. It is assumed that it will be important that the earth stations be kept continuously informed of the status of both the crew and spacecraft as well as maintain near continuous tracking contact throughout all phases of the mission. Second priority is given to gathering, storing and transmitting to earth stations of scientific information which contributes directly to the man-in-space effort. Third priority is given to instrumentation such as onboard television and terrain surveillance radar as aids to exploration for lunar landing and rendezvous with other space stations.

In compliance with the Apollo guidelines, the proposed system will make maximum use of existing and/or programed ground communications and tracking networks consistent with the requirement for adequate real-time tracking and data transmission. The design will emphasize minimum weight and input power requirements, ease of operation and maintenance, and a maximum in reliability. To achieve these objectives, miniaturization and modular construction, consisting of plug-in units of standardized dimensions with centralized cooling techniques will be used wherever feasible. Maximum effort will be made to conserve vehicle power by the use of high gain directional antennas when beyond the earth's atmosphere, by programing an increase of radiated power with distance from the earth, by time sharing, and by multiple equipment usage. The equipment will be designed for manual operation and will depend on the crew for such operational functions as navigational fix-taking, positioning of antennas, changing power levels and receiver bandwidths, switching from near earth to deep space voice telemetry and tracking data links, monitoring of equipment operation, performing in-flight maintenance and other related functions.

The Apollo project as presently conceived will extend over a period of at least 10 years. Consequently, in order to keep the equipment up with the state of the art and to facilitate meeting new requirements as the program advances, growth potential must be incorporated throughout the basic design. This is important on a project of this nature which will involve many specialized missions such as earth orbital, space

laboratory, rendezvous, and circumlunar, where the requirements can change drastically.

The feasibility of integrating radar rendezvous and altitude determination capability with the overall system will require further study. This will include a comparison of FM/CW and pulse-type radars with particular reference to antenna configurations, weight advantages, and target acquisition and tracking capabilities peculiar to each type.

Real-time television as a data link for the transmission of visual information, such as lunar terrain features or for the observation of men at work in a space environment, will be investigated as a third priority system, particularly from the standpoint of power requirements, weight and size.

Real-time radar tracking data has been found to be lacking in accuracy for trajectory analysis, especially at low sighting angles where multipathing affects cause severe fluctuations in azimuth, elevation and range errors. Nulls in the antenna patterns, misses in beacon triggering, or other reasons for momentary weak signals (20 decibels or less) also result in excessive angular and range error fluctuations. This experience indicates that the tracking requirements prior to insertion may not be fully realized with the present C-band radar, and that a more precise system is needed. For this reason, newly developed tracking devices such as Missile Trajectory Measurement System (MISTRAM), which have possibilities for more accurate real-time data, will be investigated thoroughly as a possible replacement for the present C-band radar.

## INTRODUCTION

This paper presents the results of a preliminary study of the Apollo spacecraft communications and tracking requirements. Presented also is a brief description of the subsystems proposed to furnish the detailed requirements of telemetry, two-way voice, television and beacon tracking. This preliminary study will be followed by a more comprehensive design study leading to a firm outline of the essential technical parameters and boundary conditions for incorporation into an overall specification.

## GENERAL

A communications and tracking system is proposed to:

(a) permit continuous monitoring on the ground of the status and progress of the mission, and the performance of the spacecraft equipment

(b) assure ground detection of spacecraft emergencies

(c) provide precise tracking information

(d) return scientific data at the information rates required

(e) provide information necessary to assure that recovery can be accomplished.

The system proposed will assure a continuous data link and two-way voice between the earth and spacecraft at all times except while the vehicle is on the far side of the moon. Storage and playback facilities will be provided when transmission of data in real-time is impractical.

The national missile ranges, as well as the present Mercury range, are equipped with conventional HF and UHF transmitting and receiving equipment, FM telemetry, operating on the standard telemetry band (225 to 260 mcs) and C- and S-band tracking radars, whereas the Deep Space Instrumentation Facility (DSIF) uses a sensitive FM communications and C. W. Doppler tracking system operating in the frequency band of 2,115 to 2,390 mcs. Since the number and location of the DSIF stations will not permit continuous line-of-sight tracking there will be a need for two systems for the Apollo vehicle: one for earth proximity to operate in conjunction with the existing missile ranges, and the other to be compatible with DSIF existing or programed for the 1964 to 1970 period.

For launch preinjection and near earth orbital missions, it is proposed to use equipment compatible with the missile ranges and/or the present Mercury range. For deep space (two-earth radii or more from the earth), the DSIF will be used. These facilities have been proven reliable on other space programs having similar tracking and data link requirements. The DSIF operating frequency band (2,115 to 2,295 mcs) is a particularly desirable trade-off for transmission at great distances. The most important advantages of this frequency band have been found to be:

(a) Vehicle-borne equipment in this band is within the state of the art for the Apollo time period.

(b) This frequency band has comparatively low atmospheric absorption and atmospheric noise.

(c) It permits the use of equipment with considerable growth potential.

(d) The size and weight of equipment and high gain antenna systems are a minimum over this frequency range.

## EQUIPMENT PROPOSED

In order to determine the characteristics of the communications and tracking system for the Apollo spacecraft, certain basic assumptions had to be made. The most significant of these are:

(a) Facilities of the DSIF and the national missile ranges can be made available in support of the project.

(b) Landing facilities of Edwards Air Force Base, California can be used.

(c) Telemetry data requirements will be similar to that of Project Mercury.

(d) Radio frequency allocations can be obtained within the bands recommended.

A block diagram of the system proposed is shown in figure 1; general operating characteristics and phase of mission, in which each item of equipment is to be used, are listed in table I. Briefly, the equipment proposed and its function for the various phases is summarized as follows:

(a) Launch, boost, insertion and earth orbital, and reentry -

(1) Tracking - C-band beacon in conjunction with standard missile range radar equipment as the primary system with the Minitrack system for backup. The beacon will have an added command control feature available for use during unmanned, vehicle qualification or other test missions.

(2) Two-way voice communications.

a. UHF operating in the band as the 225- to 400-mcs primary system with HF (10 to 20 mcs) as backup.

b. PCM/FM when within line of sight of deep space range stations.

(3) Telemetry - PCM/FM with Minitrack as backup.

(b) Space laboratory and lunar missions -

(1) Tracking - Primary System Deep Space Net (TRAC(E))

(2) Telemetry - PCM/FM with TRAC(E) Net

(3) Voice PCM/FM with TRAC(E) Net

(4) Television PCM/FM with TRAC(E) Net (lunar missions only)

(5) X- or KU-band surveillance radar and rendezvous (lunar missions only)

(c) Recovery -

(1) UHF - CW mode of the UHF Voice T/R

(2) HF - CW mode of the HF Voice T/R

(3) Auxiliary UHF beacon 243 mcs in liferaft which may be used as a backup.

Deep Space System.- For deep space (8,000-mile altitude and up), it is proposed to use a PCM/FM system as the primary data link with an identical system as backup as shown in Blocks A1 and A2 in figure 1. This will be a two-way system compatible with TRAC(E) for the 1964 to 1970 time period. This system will measure two angles, range, and range rate, from which position and velocity may be calculated; it will also receive telemetry data and permit the transmission and reception of voice on 2,115 and 2,295 mcs respectively. PCM/FM is selected for this phase because optimum use can be made of bandwidth limitations necessary at the transmission ranges involved (210,000 nautical miles) and because of the ease in data processing on the ground. The PCM is capable of operating at approximately 103,000 bits per second on low level signals of the order zero to 200 mv with proper signal amplification. The proposed PCM/FM system will have a bandwidth of 60 kc, a 5-foot inflatable parabolic antenna (fig. 3) which will have a gain of 30 db at 2,115 mcs. It is estimated that with an absolute receiver gain of -118 dbm, an S/N ratio of 10 db and a ground antenna gain of 51 db, satisfactory ground-to-vehicle communications can be maintained at 210,000 nautical miles with a vehicle power output of 2.5 watts. (See table II.) After taking into account the losses in the power supply, modulator and transmitter, amounting to 96 percent of the input power, it is estimated that the input power will be 50 watts for an output of 2.5 watts. The weight, excluding control panel, microphone, and headset amplifiers, is estimated at approximately 26 pounds. (See table I.)

Near earth tracking and data link equipment.- For the close-in phases of the mission (launch, boost, preinsertion or earth orbital and reentry), it is proposed to use communications and tracking equipment compatible with existing networks such as the Mercury range. There will be many instances where elements of both close-in and the deep space system will be used simultaneously. As indicated in figure 1, the proposed primary and backup PCM/FM transmitters consist of a crystal-controlled oscillator operating at 259.7 mcs. This output will be amplified to the desired level and utilized as the RF carrier. This will be modulated by the voice and telemetry subcarriers for use in conjunction with ground UHF receiving equipment on near earth phases of the mission or where the vehicle is beyond the line of sight of the DSIF. When the vehicle altitude is such that it will be within line of sight of at least one of the deep space communications and tracking stations, the 259.7 carrier, referred to in the paragraph above, will be applied to a second frequency multiplier as indicated to produce a 2,333.3-mcs carrier. This is then modulated by the voice and telemetry subcarriers amplified to the required level and applied to the vehicle antenna system for operation with the DSIF. The above frequencies (259.7 and 2,333.3 mcs) have not been assigned to this project, and are therefore tentative. The 2,225- to 2,300-mcs band has been tentatively allocated to earth-space communications, and it is assumed that the final allocation will most logically fall into this band. Conventional HF (10 to 20 mcs) and UHF (225 to 399 mcs) transceivers will be used as the backup two-way voice system during launch, boost, insertion, and reentry and will serve as the primary recovery system. The types of transmission, i.e., single side-band, suppressed carrier, and so forth, modulation characteristics and other design features will be selected on the basis of assuring maximum compatibility with military and commercial communications networks contemplated for the 1964 to 1970 time period. The HF and UHF transmitters will be equipped with a CW mode of operation as a homing aid during recovery operations and will be compatible with communications receiving equipment in military and commercial aircraft and shipping. The C-band beacon, in conjunction with ground radar such as the AN/FPS-16, will be the primary tracking system for vehicle altitudes which are less than two-earth radii. Circuit quality analysis indicates a peak beacon power output of 2,500 watts is required to insure reliable tracking. To provide for command functions during the critical phases of launch and insertion, a control selector utilizing digital techniques can be incorporated with the beacon receiver to perform on-off switching functions. This can be a subsection of the beacon which can be activated by the receiver through a decoder to provide six command channels. This function can be made consistent with and compatible with existing ground radars. As a backup to the radar tracking net, the Minitrack system will be utilized. In addition to its basic function of tracking, the Minitrack beacon will serve as an RF link for telemetering data to the ground stations. For this purpose, a telemetry coder will operate in conjunction with the beacon as indicated in the block diagram, figure 1.

Rendezvous equipment.- The rendezvous problem in space is usually thought of as taking place in three steps:

(1) The initial part of the maneuver which is required to maneuver the two vehicles in relatively close proximity and the same orbit plane

(2) Search and acquisition

(3) The terminal or homing phase wherein the two vehicles are relatively close together (within 100 nautical miles).

A suitable vehicle homing system is required for this later phase, if it is found that the navigational system cannot assure an accuracy of ±50 nautical miles in position and ±50 feet per second in velocity. The equipment considered for a manned vehicle need not be completely automatic, such as the complete target seeker with guidance computer as used in ground-to-air or air-to-air missiles, but rather a manually-operated search or tracking radar in the rendezvous vehicle operating in conjunction with a beacon in the space station. Such a system would give the range and angular relationship of the two vehicles necessary to permit homing from a distance to within several feet of each other. Optical or infrared could then be used for the actual contact or docking operation. Such a system could give a maximum range of approximately 100 nautical miles with reasonable power requirements. Extensive study will be required in order to arrive at the optimum trade-off between the use of radar and infrared techniques. The type of radar (that is CW, pulse, and so forth) to be selected and its operating frequency are dependent upon many factors such as size, complexity, power requirements and its flexibility for various applications (search, tracking surveillance, terrain clearance, and so forth). A pulse-type radar appears to be the most promising. It is inherent in C. W. Doppler radar that the transmitter and receiver must be on at all times, which gives rise to the problem of keeping the transmitted energy out of receiver except for that reflected by the target. To accomplish this, it is essential that the transmitting and receiving antennas must be physically or electrically isolated. Since the mounting and stabilization of directional antennas is a formidable problem on space vehicles, it appears that the pulse-type homing radar will be the most desirable for Apollo. It is, therefore, proposed to use an X- or KU-band (13 to 16 kmcs) pulse-type radar in the rendezvous vehicle in conjunction with interrogator-responder beacon in the space station. From the circuit quality analysis as shown in appendix I, a 10-kw peak power radar with a parabolic antenna operating in conjunction with a beacon of 100-watt peak power in the space station will give the desired signal at a maximum angle of 100 nautical miles. The characteristics of the proposed radar are:

| | |
|---|---|
| Frequency | 13,000 mcs |
| Peak power out | 10 kw |
| Pulse Repetition Frequency | 620 cps |
| Pulse width | 0.25 μs |
| Receiver sensitivity | 98.5 dbm |

This radar will be available for terrain surveillance of the lunar surface, for altitude determination and reentry backup.

Television system.- The pictorial information gathered recently from closeup lunar photography may confirm the desirability of including television for a direct view of the lunar surface from optimum ranges and transmission of such data to the earth. It may also be desirable to observe the operation of the crew. For this purpose, it is proposed to include a PCM/FM television system.

Real-time television for the circumlunar missions appears to be within the state of the art, using a directional antenna system on the vehicle and the facilities of the deep space tracking net, and utilizing a digital system with storage and comparison techniques to reduce the amount of redundant information that is present in a scene scanned by usual methods. Storage elements may also be used to reduce bandwidth. Line redundancy can be lowered, resulting in additional compression. By lowering the frame rate to 20 frames per second and the resolution to 350 lines, it can be shown that the bit rate can be reduced to a value well within the 0.32 megabit per second rate required for the Interrange Instrumentation Group (IRIG) maximum bandwidth on the 216- to 235-mcs band. As shown in the circuit quality analysis given in table II, a reasonably good quality picture can be transmitted with a 1.3-watt transmitter, a transmission bandwidth of 500 kcs and a receiver gain of -147 db.

Magnetic tape recorder.- A recorder is recommended for the storage of data where real-time transmission to the earth is not practical. The type of recorder selected will be dependent upon tape size, speed, capacity, amount of data, its electrical and mechanical characteristics with due regard to reliability, size and weight.

MISTRAM.- The MISTRAM system is being developed by the General Electric Company under U.S. Air Force contract for installation at the Atlantic Missile Range (AMR) and is scheduled to become operational in May 1962. This is essentially a long baseline interferometer system designed for measuring range and range rates with an acquisition accuracy of $0.001^{\circ}$ at lunar distances. The system uses five linearly-polarized antennas whose feed systems may be rotated. It tracks the spacecraft polarization vector and positions the feeds for a maximum received signal throughout the flight. Two carriers are transmitted from the ground station to the transponder in the vehicle. One carrier is a fixed frequency, 8,148 mcs accurately stabilized to one part in $10^8$. The other is a signal swept in frequency $7{,}888 \pm 4$ mcs to provide a calibration of unambiguous range. The transponder beacon uses a Klystron oscillator which is locked to each signal radiated from the ground. The transponder provides a coherent offset and retransmits the signal to earth. It is anticipated that the present design would afford a tracking capability to the moon with a positional accuracy of 5 nautical miles if a high gain X-band paraboloid antenna is used on the vehicles. The system also has possibilities for providing telemetry and command functions on a separate carrier. Since the system appears to have extraordinary accuracy, its development will be followed for possible future integration in the Apollo system, and if found satisfactory, it should be adopted in lieu of the C-band radar for ground tracking during launch, preinsertion and near earth orbital.

## VEHICLE ANTENNA SYSTEMS

In order to provide continuous coverage with a minimum number of antennas and to maintain vehicle transmitter power requirements within reason, the use of flush-mounted omnidirectional antennas while operating in the earth's atmosphere and high-gain directional systems while in deep space will be required. The following eight antenna systems are proposed:

(a) Parabola (inflatable) - 5-foot diameter

(b) Parabola (X- or KU-band) - 1-foot diameter

(c) Three C-band beacon flush-mounted helices

(d) Discone antenna, HF, VHF and UHF

(e) Probe, 2,115 mcs

(f) Probe, 2,295 mcs

(g) HF telescoping whip

(h) UHF telescoping whip

High gain parabolic antennas are essential for deep space voice, telemetering and television. It is recommended that parabolic antennas referred to in (a) and (b) above be of the inflatable cartridge-activated types utilizing foaming materials which harden at a rapid rate to prevent deflation or damage by micrometeorites. The directional antennas must be steerable and controllable so as to keep them looking at the earth at all times they are in use.

It is not anticipated that all of the above antennas will be required for each flight. There will be certain specialized missions which will not require directional high gain antennas. The two parabolic antennas listed in (a) and (b) above will be installed on missions where the distance from the earth is two-earth radii or greater and a rendezvous operation is involved. These antenna systems will require precise stabilization to within $\pm 2^{\circ}$ for communications and data link and $\pm 0.2^{\circ}$ for deep space tracking and rendezvous if the high gain antennas are to be used for tracking.

The KU-band parabola will be installed for rendezvous missions only and will require mounting on the vehicle centerline at the forward end as shown in figure 3. This antenna will have a conical scan of $60^{\circ}$. The 5-foot parabola will be utilized for communications and data link in conjunction with the deep space tracking net and will require stabilization to within $\pm 2^{\circ}$. In order to insure proper orientation of the antenna with respect to the earth during all phases of the mission, the antenna directivity should be adjustable over the range of $180^{\circ}$ as shown in figure 3.

The C-band beacon antenna system consists of three flush-mounted cavity helices symmetrically arranged around the vehicle approximately 18 feet from the forward end. Power will be applied to each radiator by coaxial cable through a three-way power divider. The resultant radiation pattern will be circularly polarized and should be omnidirectional without without excessive nulls at a perpendicular to the longitudinal axis of the vehicle.

The discone antenna located immediately ahead of the pressurized area can be designed into the capsule structure. This configuration will act as a center-fed dipole at HF frequencies and as a discone antenna at VHF and UHF. The radiation pattern will be omnidirectional with a nominal 50 ohm impedance covering the range 20 to 400 mcs. This antenna system will be used during launch, near earth orbital and reentry phases of the mission. Two probes, one tuned to 2,115 mcs and the other to 2,295 mcs will be mounted as indicated in figure 3 for deep space and near earth tracking and as a backup to the 5-foot parabola. These will provide omnidirectional coverage and will serve as a point source for precise tracking.

The UHF recovery antenna will be a telescoping whip extended to the operating position after jettison of the discone antenna. It will provide a omnidirectional vertically-polarized pattern and will be used during the postlanding phase operating at UHF voice, unmodulated CW and telemetry frequencies.

The HF recovery antenna will also be a telescoping whip which may be cartridge-activated to the operating position on touchdown. It will provide omnidirectional coverage at HF voice and unmodulated CW during recovery operations.

## PACKAGING OF ELECTRONIC COMPONENTS

The use of a large number of individual black boxes for the various communications functions, as in Mercury, where the units are installed dependent upon the availability of space, has resulted in a rather disorganized system which has greatly complicated vehicle cabling, system cooling, system test and checkout, and has made repairs or replacement difficult and time consuming.

Based on this experience, modular packaging and integration of as much of the electronic equipment as is feasible in modular form, as illustrated in figure 2, is recommended for the Apollo project. This will permit the removal of an entire group of modules for intersystem checkout and adjustment with minimum dislocation and handling of vehicle cabling. This concept of packaging electronic components has been in use for some time in aircraft and missiles in such applications as voice, command, telemetry, analog and digital computers, and so forth. The subassemblies consist of a series of modules of standardized dimensions packaged as shown in figure 2. The optimum size of these units appear to be approximately $3\frac{1}{2}$-inches wide by $2\frac{1}{2}$-inches high with the length variable to accommodate subsystems of varying complexity and size. The optimum dimensions can be more accurately determined after the system has been breadboarded in the laboratory.

The use of modular integrated packaging has been accepted with good results in high performance military aircraft and missiles. The major advantages are:

(a) Reduces the spacecraft interconnecting cabling which becomes exceedingly complex and space consuming where the individual black box approach is used.

(b) Simplifies the equipment cooling problem in that the high heat-producing units can be systematically arranged to permit the use of a highly centralized and efficient cooling system where the cooling parameters can be accurately determined and controlled. (See fig. 2.)

(c) Permits a reduction in overall system weight of approximately 30 to 40 percent.

(d) Simplifies design and permits ease of upgrading or making design changes as the program advances. The use of standardized modules for amplifiers (audio, intermediate frequency (IF) and so forth), power supply subassemblies, and so forth, permits repetitive use.

(e) Makes possible in-flight maintenance and minimizes the requirement of redundancy for reliability. The only in-flight maintenance will be the replacement of individual plug-in units.

The micromodular construction for packaging electronic components as currently being developed by Radio Corporation of America (RCA) under U.S. Army Signal Corps contract and the thermally integrated micromodule technique (TIMMS) as developed by the General Electric Company appear to have possibilities of size and weight reductions ranging from 50 to 90 percent and should be available for production within the next 2 or 3 years. The progress of these developments should be followed closely to determine what aspects of this technique can be incorporated in the packaging of electronic equipment for the Apollo vehicle.

The weight and size estimates given in table II are based on the present state of the art. Utilizing micromodular techniques should result in further weight reductions.

## HEAT DISSIPATED BY ELECTRONIC EQUIPMENT

In a space vehicle, the removal of heat dissipated by the onboard equipment presents a major problem. The development of excessive temperatures and hot spots in electronic equipment results in lower reliability and degraded performance. For this reason the most efficient and effective cooling system, is recommended, such as the system shown in figure 1, where such parameters as the amount of heat to be removed, cooling media, and temperatures can be more accurately controlled. An estimate of the heat dissipated by each subsystem of the communications and tracking equipment is included in table III. It is estimated that the total heat to be removed for a lunar mission of 14 days will be of the order of 12,300 watt-hours.

## RELIABILITY

An attempt was made in Project Mercury to assure reliability through redundancy. It is conceivable that if redundancy is carried too far, it can generate insurmountable maintenance problems as well as add excessively to the time for checkout and replacement of defective units, resulting in a system with such a low operational capability that the reliability gained becomes questionable. It appears that more can be gained by the use of an integrated system where adequate equipment cooling can be controlled employing plug-in modules which will facilitate trouble location and quick replacement of defective units during both ground checkout and while in flight. For this reason, an integrated system utilizing individual plug-in units is recommmended for Apollo, with the only duplication being in the use of two identical PCM/FM systems contained in separately assembled packages as shown in figure 2, so that if one failed during a critical phase of the mission, the other would be available for immediate operation.

## OPERATIONAL PLAN

A schematic of the operational concept for a typical Apollo mission is shown in figure 4. In the proposed plan, the vehicle is launched at Cape Canaveral, Florida utilizing the facilities of AMR and the Mercury range for the launch to injection phases of the mission. After the vehicle is in orbit and has gained sufficient altitude to be visible at all times from at least one of the three DSIF stations, control will be transferred to this net.

During midcourse phase of the mission, the tracking system will utilize the DSIF network to obtain data for computation of trajectory, midcourse corrections, and antenna pointing information. Since communications satellite relay systems capable of meeting a variety of military and civilian applications may become operational during the Apollo time period, studies should be made of the possibility of serving as relay stations for part of the vehicle communications links. For reentry and landing at Edwards Air Force Base, use could be made of the early warning, tracking and control facilities of the Western Air Defense Command and the Pacific Missile Range (PMR) for the approach to the designated landing site. In each instance, regardless of the tracking network used, the tracking data are transmitted from the tracking stations to the computing center at NASA Goddard Space Flight Center (GSFC) for processing in the digital computer. Since a large amount of data will be received, some statistical method must be used for smoothing purposes.

# APPENDIX I

## X-Band Radar Circuit Quality Analysis

The radar parameters are used in calculating the circuit margin for the X-band homing radar and interrogator-responder beacon at 100 nautical miles range. The two-way voice and one-way telemetry data links were calculated in a similar manner and the results are shown in table II.

(a) Radar-beacon link - X-band (13,000 mcs)

| | |
|---|---|
| Radar peak power out | 40 dbw 10 kw |
| Radar RF loss | -4 db |
| Radar antenna gain (1-ft parabola) | 30 db |
| Wave guide loss in rendezvous vehicle | -4 |
| Space loss | $-37 + 20 \log f + 20 \log d = -160$ db |
| Beacon antenna gain | 10 db |
| Beacon line loss | 2 db |
| Power at beacon receiver in Space Station | $= 62 - 160 + 10 = -90$ dbw |

(b) Beacon radar link - X-band (13,000 mcs)

| | |
|---|---|
| Beacon peak power out | 20 dbw (100 w) |
| Beacon RF loss | -1 db |
| Radar antenna gain | +30 db |
| Wave guide loss | -1 db |
| Space loss | $-37 + 20 \log f + 20 \log d = 160$ db |
| Beacon antenna gain | 10 db |
| Power at radar receiver | $48 - 160 + 10 = -110$ dbw |

TABLE I.- PRELIMINARY STUDY OF APOLLO SPACECRAFT COMMUNICATIONS

AND TRACKING SUBSYSTEM CHARACTERISTICS

| Equipment name and function | Xmtr pwr (watts) | Rcvr. sens. | Mod. and BW | Input pwr (watts) | Wt. (lbs) | Volume (ins.) W. H. L. | Phase of mission in which equipment is used |
|---|---|---|---|---|---|---|---|
| 1. UHF voice T/R 225 to 399 mcs | 1 | 2.5µv | A 6 kcs | 11 | 3.0 | $3 \times 2\frac{1}{2} \times 4$ | Launch, insertion, near earth orbital. Provide unmodulated 1,000 cps tone during recovery phase. |
| 2. HF voice T/R 10 to 20 mcs | 1 | 2.5µv | A 6 kcs | 12 | 3.0 | $3 \times 2\frac{1}{2} \times 7$ | Launch, injection, near earth orbital. Unmodulated 1,000 cycle tone during rescue. |
| 3. Minitrack and T/M coder | 0.25 | N/A | A | 3 | 0.5 | $3 \times 2\frac{1}{2} \times 1$ | Near earth tracking and telemetry data on 10.5 kc modulation. |
| 4. C-band beacon | 2,500 | -95 dbw | P 8 mc | 40 | 13 | $3 \times 2\frac{1}{2} \times 24$ | Launch, injection, near earth orbital, reentry tracking. Accept command signals. |
| 5. T/M and voice comm. 2115 to 2390 mcs | 2.5 | N/A | 160 kc PCM/FM | 50 40 | 26 | $6 \times 2\frac{1}{2} \times 24$ | Compatible with TRAC(E) deep space system. |
| 6. TV system 2115 to 2390 mcs | 1.3 | N/A | PCM/FM 500 kcs | 75 | 66 | 2,300 cu. in. | Lunar missions only. |

TABLE I.- Continued

| Equipment name and function | Xmtr pwr (watts) | Rcvr. sens. | Mod. and BW | Input pwr (watts) | Wt. (lbs) | Volume (ins.) W. H. L. | Phase of mission in which equipment is used |
|---|---|---|---|---|---|---|---|
| 7. Power amp | | | | | | | Used only for ranges beyond 700 mm. To increase power output of HF T/R and UHF T/R. |
| HF (10 to 20 mcs) | 4 | - | 6 kcs | 16 | 1.5 | $3 \times 2\frac{1}{2} \times 2$ | |
| UHF (225 to 399 mcs) | 4 | - | 6 kcs | 15 | 1.5 | $3 \times 2\frac{1}{2} \times 2$. | |
| 8. Voice-data multiplexer | - | - | - | - | 3.0 | $3 \times 2\frac{1}{2} \times 6$ | Provide audio mixing. |
| 9. Control Center | - | - | - | 1.0 | 1.5 | $5\frac{3}{4} \times 3 \times 2$ | Rcvr. vol controls, Xmtr switches. |
| 10. Multiplexer (HF, VHF and UHF) | - | - | - | - | 6.0 | $5\frac{3}{4} \times 8 \times 3$ | Multiplexing HF, UHF and VHF frequencies. |
| 11. Antenna Systems | | | | | | | |
| Deep space | - | - | - | - | 16.0 | 18 × 5 dia. | Use an inflatable parabola which is stowed during launch, jettisoned at reentry. |
| X-band | - | - | - | - | 8.0 | 10 × 5 dia. | |
| Orbital launch HF, VHF and UHF | - | - | - | - | 12.0 | | |
| C-band | - | - | - | - | 6 | 8 × 6 × 4 | Flush-mounted helix. |
| UHF Recovery | - | - | - | - | 1.0 | 6 × 1 (stowed) | Telescoping whip antennas, which are extended to the operating position at impact. |
| HF Recovery | - | - | - | - | 3.0 | 15 × 3 dia. | |

TABLE I.- Concluded

| Equipment name and function | Xmtr pwr (watts) | Rcvr. sens. | Mod. and BW | Input pwr (watts) | Wt. (lbs) | Volume (ins.) W. H. L. | Phase of mission in which equipment is used |
|---|---|---|---|---|---|---|---|
| 12. X-band alt. determination and homing radar | 2.5 kw | -120 dbm | Pulse 0.25μs | 450 | 50 | 600 cu. in. | For lunar missions and homing for space rendezvous. |
| 13. X-band plumbing | - | - | - | - | 8 | | From R-T unit to X-band antenna. |
| TOTAL | | | | | 230 | 8,330 (cu. in.) | |

TABLE II.- APOLLO SPACECRAFT ESTIMATED CIRCUIT QUALITY ANALYSIS

COMMUNICATIONS AND TRACKING SUBSYSTEMS

| Circuit | Direction, vehicle ref. | Freq. mcs *** | Power Xmitted | | Line loss | $G_T$ db | Distance nautical miles** | Free space loss db | $G_R$ db | $L_{CR}$ db | $P_R$ dbw | Rcvr. sens. dbw | Actual margin |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | watts | dbw | | | | | | | | | |
| UHF Voice T/R with power amplifier | from | 240 to 399 | 5.0 | 7 | 3 | -0 | 8,000 | 162 | 18 | 1 | -141 | -154 | 13 |
| | to | 240 to 399 | 100 | 20 | 1 | 18 | 8,000 | 162 | -3 | 0 | -128 | -142 | 14 |
| HF Voice T/R with power amplifier* | from | 10 to 20 | 5.0 | 7 | 1 | -3 | n/a | | 6 | 1 | | -124 | n/a |
| | to | 10 to 20 | 100 | 20 | 1 | 6 | | | -3 | 3 | | -124 | |
| Minitrack | from | 135 | 0.200 | -7 | 1 | 6 | 8,000 | 156 | 20 | 0 | -138 | -143 | 5 |
| Microlock | from | 108 | 0.200 | -7 | 1 | 6 | 8,000 | 154 | 18 | 0 | -138 | -154 | 16 |
| C-band beacon | from | 5,400 | 2,500 | 34 | 2 | 0 | 8,000 | 190 | 44 | 2.0 | -116 | -130 | 14 |
| | to | | 3 mw | 95 | 2 | 44 | 8,000 | 190 | 0 | 2.0 | -55 | -95 | 40 |
| Telemetry** | from | 2,200 | 2.5 | 4 | 4 | 28 | 210,000 | 210 | 51 | 2 | -134 | -139 | 6 |
| TV transmitter** | from | 2,200 | 1.3 | 1.1 | 6 | 29 | 210,000 | 210 | 51 | 2 | -137.4 | -147 | 10 |
| Command section of C-band receiver | to | 5,500 | 5 mw | 67 | 1 | 40 | 8,000 | 190 | -4 | 0 | -88 | -95 | 7 |

TABLE II.- Concluded

| Circuit | Direction, vehicle ref. | Freq. mcs *** | Power Xmitted | | Line loss | $G_T$ db | Distance nautical miles** | Free space loss db | $G_R$ db | $L_{CR}$ db | $P_R$ dbw | Rcvr. sens. dbw | Actual margin |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | watts | dbw | | | | | | | | | |
| X-band surveillance and homing radar | from | 13,000 | 10 kw | 40 | 8 | 30 | 100 | -166 | 30 | 2 | -76 | -95 | 19 |
| | to | | 100 | 20 | 2 | 30 | 100 | -166 | 30 | 8 | -96 | -110 | 14 |
| Voice PCM/FM | from | 2,200 | 2.5 | 4 | 4 | 30 | 210,000 | 210 | 51 | 2 | -132 | -139 | 7 |
| DSIF** | to | 2,115 | 100 kw | 50 | 2 | 51 | 210,000 | 210 | 28 | 4 | -87 | -110 | 23 |

* Maximum range of HF transmission is dependent upon the ionosphere and to lesser extent upon equipment design.

** The distance from the earth to the moon is considered to be the mean distance of 210,000 nautical miles.

*** Operating frequencies have not yet been assigned. The bands proposed appear to be optimum for space communications in the Apollo time period.

TABLE III.- APOLLO SPACECRAFT ESTIMATED INPUT POWER

AND HEAT DISSIPATION 14-DAY MISSION (336 HOURS)

COMMUNICATIONS AND TRACKING SYSTEM

| Phase of Mission | Duration hrs | Duty Cycle | | Input Power | | Power Dissipated | |
|---|---|---|---|---|---|---|---|
| | | % | hrs | watts | watt-hrs | watts | watt-hrs |
| Launch, injection and reentry | 2.0 | | | | | | |
| Voice** | | 50 | 1.0 | 24 | 24 | 22 | 22 |
| Telemetry | | 100 | 2.0 | 3 | 6 | 2.5 | 5 |
| Tracking, C-band | | 100 | 2.0 | 40 | 80 | 38 | 76 |
| Command | | 100 | 2.0 | 1.25 | 2.5 | 1.0 | 2 |
| Midcourse | 118 | | | | | | |
| Voice | | 12 | 15 | 50 | 750 | 48 | 680 |
| Telemetry | | 50 | 60 | 50 | 3,000 | 48 | 2,840 |
| X-band radar | | 2 | 3 | 450 | 1,350 | 400 | 880 |
| Tracking TRAC(E) | | 90 | 105 | 3 | 315 | 2 | 210 |
| Data storage | | 50 | 59 | 5 | 295 | 4.8 | 290 |
| Lunar Orbital | 216 | | | | | | |
| Voice | | 9 | 20 | 50 | 1,000 | 48 | 820 |
| Telemetry | | 23 | 50 | 50 | 2,500 | 48 | 2,400 |
| Television | | 7.5 | 16 | 90 | 1,440 | 88 | 1,400 |
| Tracking TRAC(E) | | 50.0 | 108 | 3 | 324 | 2 | 216 |
| X-band radar | | 1.5 | 3 | 450 | 1,350 | 400 | 1,280 |
| Data storage | | 50 | 108 | 5 | 540 | 4.8 | 518 |
| Recovery** | 72 | | | | | | |
| HF T/R | | 50 | 36 | 12 | 432 | 10 | 360 |
| UHF T/R | | 100 | 72 | 12 | 864 | 10 | 720 |
| Receiving* | 336 | 100 | 336 | 1.25 | 420 | 1 | 336 |
| Total | 408 | - | - | - | 14,446 | - | 13,055 |

*It is assumed that two voice receivers are on at all times from launch to recovery.

**Voice communications over UHF with HF as backup assumed during this phase.

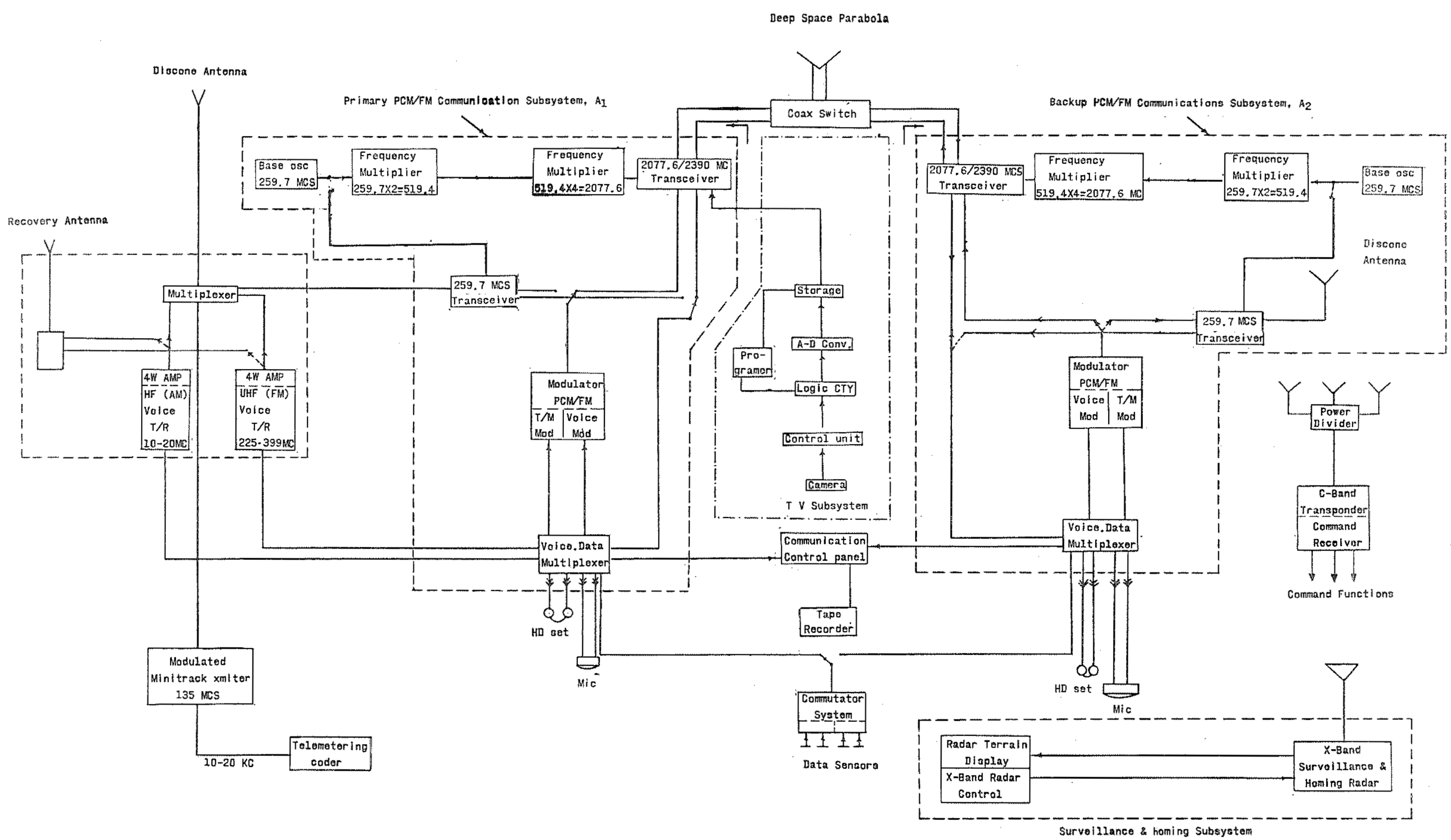


Figure 1.- Proposed spacecraft electronic system.

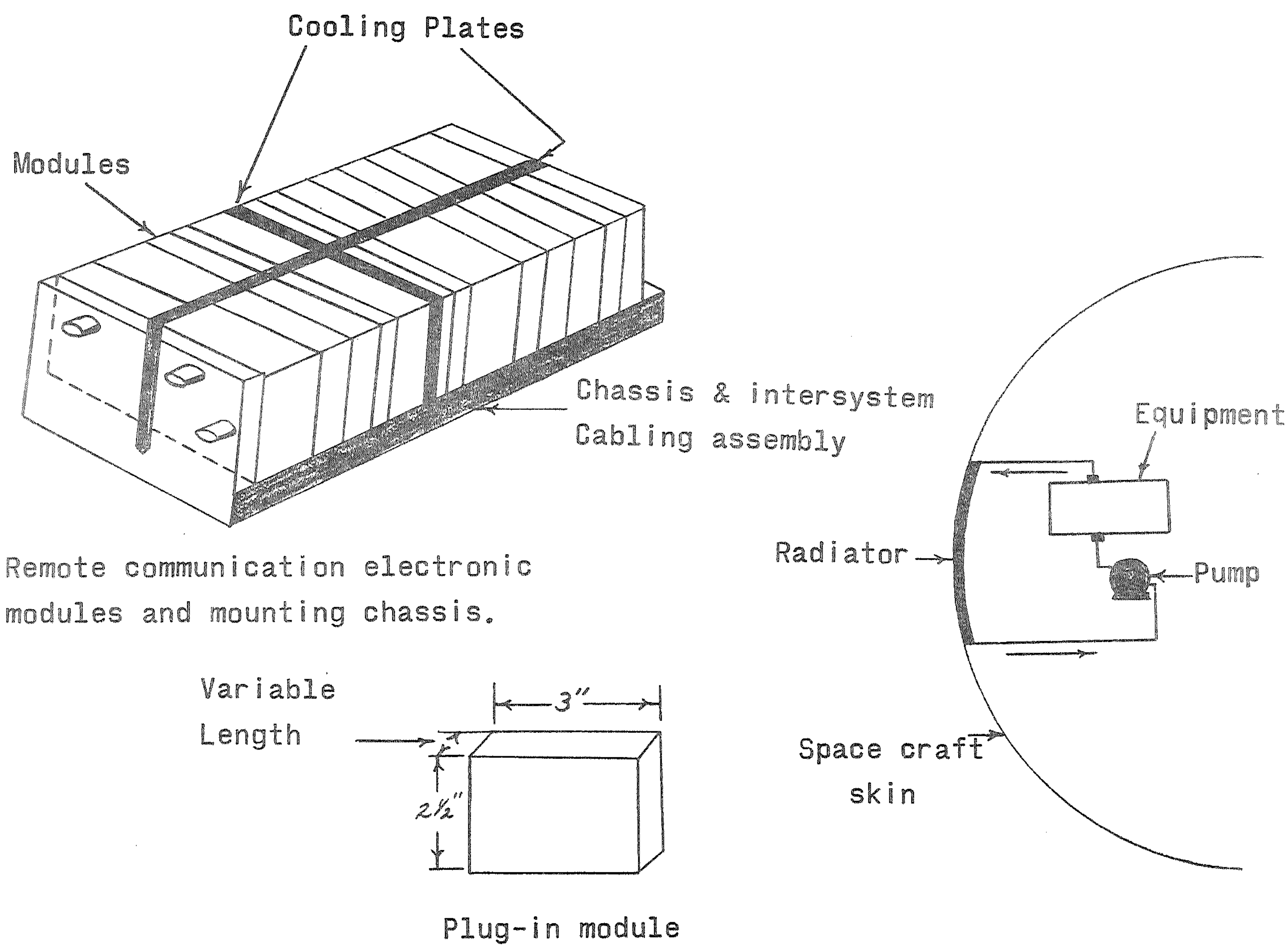


Figure 2.- Equipment cooling system schematic.

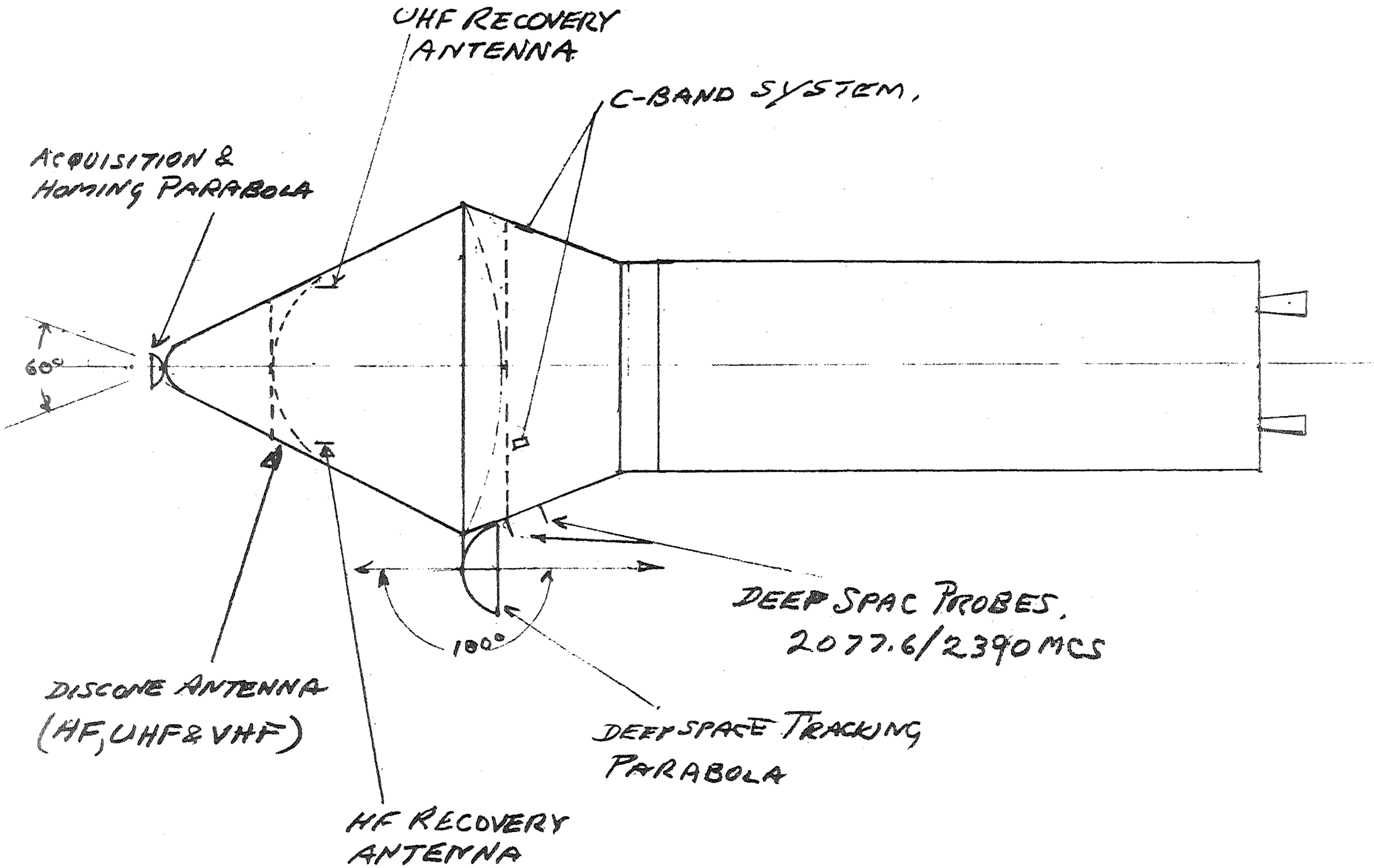


Figure 3.- Proposed vehicle antenna systems.

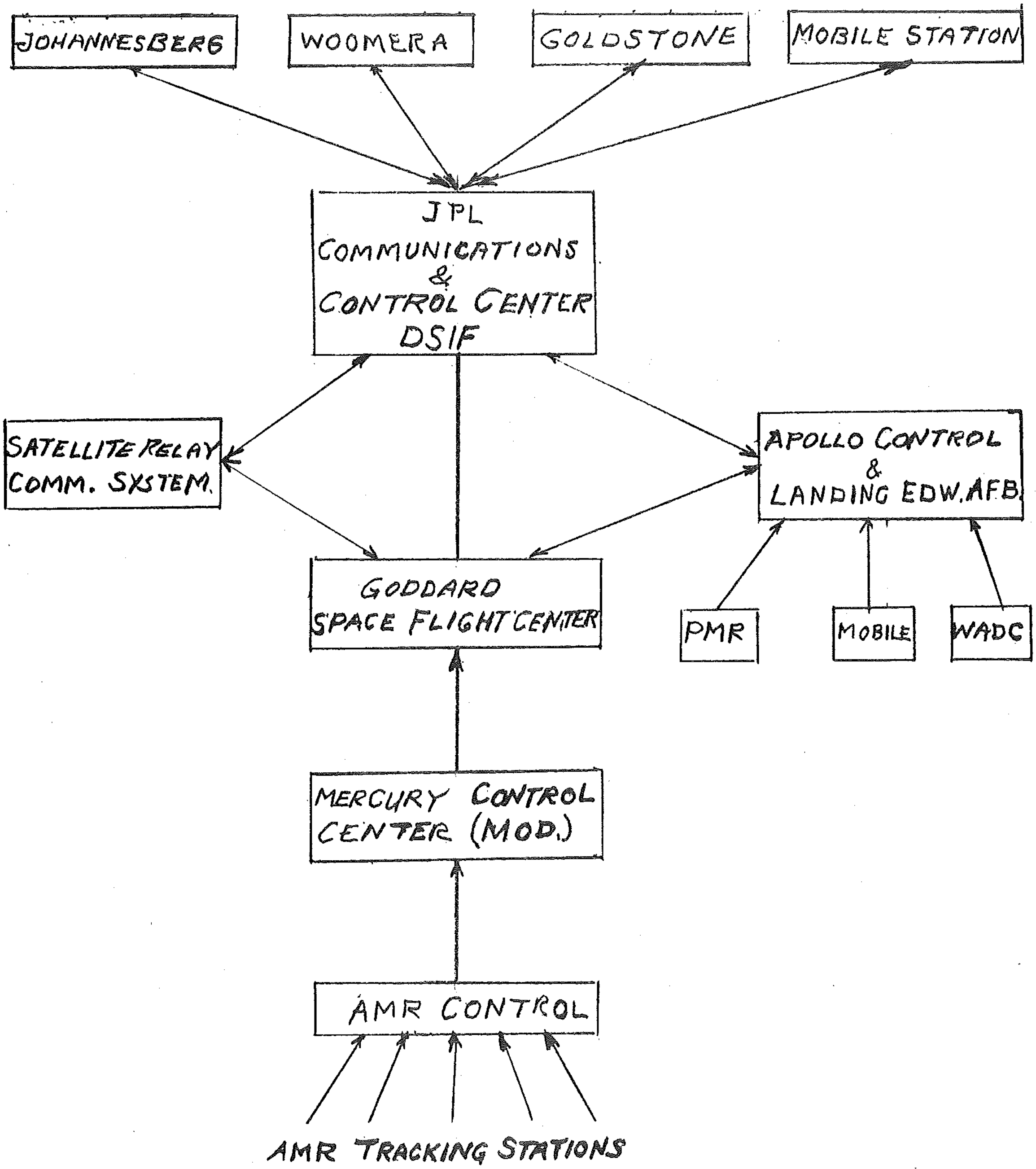


Figure 4.- Operational plan - ground communications and tracking net.